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SURVEY OF FUTURE REQUIREMENTS FOR LARGE SPACE STRUCTURES

John M. Hedgepeth

Prepared by

ASTRO RESEARCH CORPORATION

Santa Barbara, Calif.

for Langley Research Center



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16. Abstract <p>The paper examines the likely future requirements for large space structures and provides the foundation for long range planning of technology development for such structures. Attention is concentrated on a period after 1985 for actual use. Basic ground rule of the study was that applications be of significant importance and have promise of direct economic benefit to mankind. The inputs to the study came from visits to a large number of government and industrial organizations, written studies in current literature, and approximate analyses of potential applications. The paper identifies diverse space applications for large area structures in three general categories: (1) large surfaces for power, (2) large antenna to receive and transmit energy over the radio frequency bandwidth, and (3) space platforms to provide area for general utilizations.</p>					
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SURVEY OF FUTURE REQUIREMENTS

FOR LARGE SPACE STRUCTURES

By John M. Hedgepeth
Astro Research Corporation

INTRODUCTION

Large deployable structures that can be launched in a packaged condition and expanded in space have been a necessary part of space flight since the very beginnings. Hundreds of furlable booms with lengths up to hundreds of meters have been flown. Hinged-arm assemblies are commonplace, and folding solar arrays are used on many spacecraft. The deployed meteor-panel arrays on Pegasus and the solar-power array on Skylab are examples of large areas. The Echo spacecraft, deployed by inflation, was certainly outstandingly large for its time. The 10-m-diameter antenna on ATS-6 is the most recent example of successful establishment of a large deployable structure with impressive performance.

Future space-flight programs will, of course, continue to require large deployable structures. The Shuttle will bring to space flight an increased frequency and reduced cost of orbital operation and an expanded universe of operational possibilities. This expansion will have to be supported by an appropriate advancement in the capability and performance of large deployable structures.

The purpose of this paper is to examine the likely future requirements for large space structures in order to furnish the foundation for the long-range planning of technology development of such configurations. Inasmuch as attention is concentrated on the period after 1985 for actual use, the examination is subject to all of the difficulties of forecasting events more than a decade away. However, enough information exists to identify the type and rough orders of magnitude of the required structural applications, at least well enough to guide the early phases of the technology development program.

Some limitations and assumptions are necessary in a study of this type. A basic ground rule of the study is that the applications be of significant importance in order that the ensuing technology development be directed along clearly useful lines.

In the present instance the decision was made to confine attention to those missions which promise a direct economic benefit to mankind. Science per se is not insignificant, but there is little enthusiasm for the support of technology development programs aimed at purely scientific endeavors. On the other hand, there is a growing conviction that space flight will be of direct economic benefit, and to such an extent to produce a blossoming of space utilization in the late 1980s.

There is one application which would place a demand on large deployable structures technology that far surpasses any other expectable demand. That is the space solar power system concept first brought forward by Peter Glaser and subsequently studied by NASA with a team of contractors (Ref. 1). Studies of this approach to helping the earth's energy problem are continuing. The main question being investigated is that of economic feasibility. If these studies arrive at a positive conclusion, the space solar power system will undoubtedly receive a great deal of attention. Indeed, the requirements of this system for structural technology would assume a commanding importance over the next decades. Inasmuch as this particular application is receiving such a large amount of detailed attention, it has been specifically excluded from the present study.

As mentioned previously, extendible booms and other "one-dimensional" structures have found a great deal of use in the past and will be employed in many future space programs. The objectives of the present study are to treat significant advances in technology and technological requirements. The one-dimensional structures, although they have a number of significant problems as future applications are envisioned, do not require the degree of technological advancement at which the present study is aimed. These one-dimensional structures are therefore excluded from the study.

The body of this paper contains the findings of the survey study together with an outline of the bases for these findings. Detailed analyses and associated references for several of the surveyed areas are given in the appendixes.

APPROACH

Three types of activities were used in the study. First, visits were made to a large number of government and industrial organizations. In these visits the objective of the study was presented, and personnel of the visited organizations were asked to come forward with studies, ideas, analyses, concepts, or any other information that would be pertinent to the future requirements of large space structures. During these discussions the attempt was made to free the conversations of concern as to whether a certain approach would be technically "practical". Indeed, people were asked to assume that a power system with a power-mass efficiency of 2000 W/kg (about two orders of magnitude beyond current flight technology) could be attainable. They were also asked to assume that a 1-km-diameter antenna could be erected in space for use at X-band (three orders of magnitude better than current technology). The response to this type of information gathering was largely favorable. Once the persons visited really became convinced that they were being asked to cast their minds adrift from "practicality" and to operate with the constraint of only the laws of physics, some lively interchanges resulted. It is believed that these visits yielded a fairly complete picture of current thinking about future space-flight operations requiring large structures. A list of the organizations visited is given below.

Aerospace Corporation, El Segundo, California

COMSAT, Washington, D.C.

Convair Aerospace Division, General Dynamics Corporation,
San Diego, California

General Electric Company, Valley Forge, Pennsylvania

Grumman Aerospace Corporation, Bethpage, New York

Lockheed Missiles and Space Company, Sunnyvale, California

McDonnell Douglas Corporation, Huntington Beach, California

Rockwell International, Seal Beach, California

TRW, Redondo Beach, California

NASA, Goddard Space Flight Center, Greenbelt, Maryland

NASA, Headquarters: -OA, -OAST, -OMSF, Washington, D.C.

NASA, Lewis Research Center, Cleveland, Ohio

NASA, Marshall Space Flight Center, Huntsville, Alabama

Jet Propulsion Laboratory, Pasadena, California

The second type of activity was the perusal of a number of written studies, some of which were obtained during the visits, but most of which was obtained through library searches, augmented by suggestions from a number of additional persons who work in the field. This literature study rounded out the information gathered during the visits.

The third activity was to carry out approximate analyses of some of the applications with the intent of establishing order-of-magnitude estimates of feasibility, technical payoff, sizes, and other significant parameters. These approximate investigations are referred to herein along with the significant findings of the other activities.

CURRENT AND NEAR-FUTURE APPLICATIONS

Deployable structures are being used to a considerable extent in the present space program. It is of interest to examine the current and near-future applications that are advanced since these define the forefront of the state of the art. Attention is directed to the applications involving large surfaces summarized in Figure 1. Here the "size" (the square root of the frontal area) is plotted against mass per unit area.

The applications fall into three general classifications: antennas, power supplies, and other varied uses. The last of these classifications is exemplified by deployable structures such as the 100-ft-diameter Echo satellite which was erected by inflation and, once inflated, was maintained in its elliptical shape by shell stiffness. Another example is the Pegasus satellite in which large meteoroid-detection panels were hinged out in a zig-zag fashion. A final example is a NASA LeRC radiator design which rejects heat from a satellite nuclear-power system. The radiator is a 357-m² cylindrical assemblage of beryllium pipes representative of current advanced design concepts.

The antenna classification has numerous representatives. The most advanced space probes are Mariner 10 with a 1.4-m-diameter antenna, and Pioneers 10 and 11, with a 2.7-m-diameter S-band communication dish. Geosynchronous orbiting communication satellites are currently utilizing antennas, not shown in Figure 1, with diameters up to 1.5 m at C-band and 1.1 m at X-band (of course, the higher the frequency the more difficult the achievement of large apertures). None of these antennas is deployable although some are rotated into operational position after launch.

The first truly deployable antenna is the 9.2-m S-band dish on the ATS-6 satellite. The achievement of good accuracy on this large a dish is a marked step forward in antenna technology; this technology continues to be advanced. Various companies have proposed methods of constructing larger, more accurate antennas, although such designs are not shown in Figure 1. Among these are Lockheed (based on ATS-6 technology), Convair (Warren-truss back-up structure with a suspension-bridge type of support for the reflector surface), Radiation, Inc. (ribbed back-up structure with a dual surface reflector), Grumman (flat or conical lens surfaces rather than reflectors), and Astro (spin-deployed and stiffened reflector dish). Among the Government agencies, JPL appears to be the most active in developing technology for large-aperture antennas.

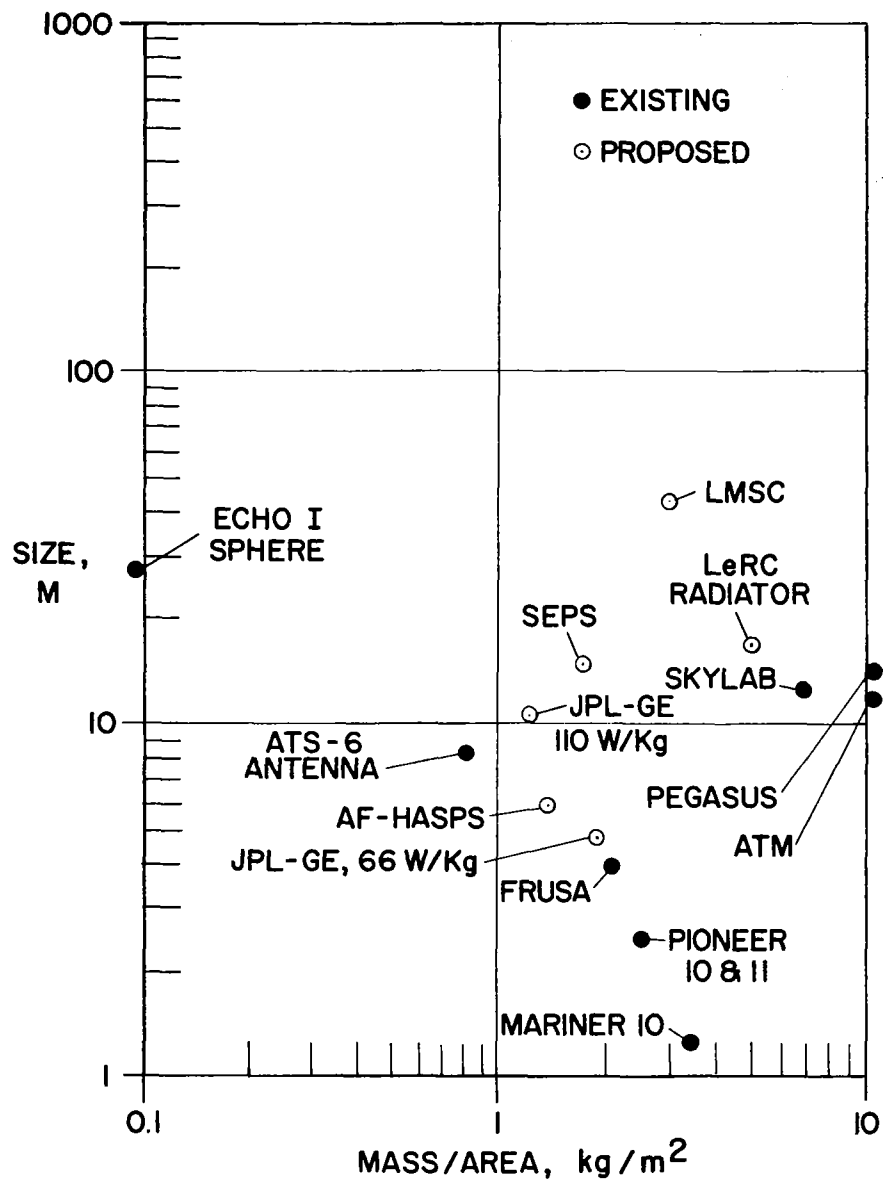


Figure 1. Current Large-Area Space Structures

Their favored approach uses a line feed together with a conical, rather than paraboloidal, reflector. A conical surface is more readily packaged and accurately deployed than is a doubly curved surface.

In the area of power generation, Skylab was the first to use deployable configurations more complicated than simple paddles. Both the "wings" on Skylab itself and the solar panels on the Apollo Telescope Mount (ATM) are deployed and packaged by an expandable lattice structure. In an effort to reduce weight, the Air Force and Hughes developed FRUSA, which is a roll-out solar array. That design is currently being improved under the HASPS program. JPL has worked with General Electric to develop engineering models of 66- and, more recently, 110-W/kg roll-out solar arrays. Fold-out solar arrays with flexible substrates were proposed and investigated to the point of experimental hardware by Lockheed (LMSC) for NASA, Johnson Space Center. This type of technology is being employed on a Canadian communications test satellite and is being recommended for the power system for the Solar Electric Propulsion System (SEPS).

An informative indicator of the uses for deployable structures in the next ten years is provided from the study of Shuttle missions and payloads that Convair is doing for NASA, Marshall Space Flight Center. Tables I through III were prepared by Convair personnel as a "deployable structures" distillation of the voluminous information derived from their studies. An examination of these tables shows that most of the requirements are within the present state of the art. This is to be expected inasmuch as mission proposers have a tendency to avoid systems which require an obvious advance in technology lest their proposal be down rated. It is interesting that a few applications, such as the 30- by 35-m Shuttle Imaging Antenna A mentioned in Table I, do require a significant advance in the state of the art.

TABLE I. LARGE ANTENNA STRUCTURES

<u>Payloads or Missions</u>	<u>Weight (kg)</u>	<u>Deployed Size (m)</u>	<u>Remarks</u>
Disaster Warn Sat Antenna	59	5.8 D	Not retrieved
Shuttle Imaging Antenna A	1424	30 x 35	L- and C-band array
Microwave Antenna B	1424	10 to 15	S- through K _a -band dish
Advanced Techn Lab, Micro Rad	136	3.8 D x 7.6	Retractable horn
Advanced Techn Lab, Micro Antenna	1360	0.15 x 2 x 25	Folded slotted waveguide
Comm/Nav SORTIE Reflector	3000*	30 D	Retrieve

*Includes erection mechanism

TABLE II. SOLAR-CELL ARRAYS -
DEPLOYED AUTOMATED PAYLOADS

<u>Payload</u>	<u>Array Size (m²)</u>	<u>Number Required</u>	<u>Deployment Concept</u>
LST	7.99	4	Two pair roll out
Lyman Alpha Expl	1.80	2	Rigid panel, folded
Cosmic Background Expl	1.80	2	Rigid panel, folded
Adv Radio Astron Sat	1.80	2	Rigid panel, folded
Large X-ray Rac	6.97	4	Two pair roll out
Extended X-ray Surv	6.97	4	Two pair roll out
Small Hi-Energy Obs	1.80	2	Rigid panel, folded
Large Hi-Energy Obs A	4.64	4	Two pair roll out
Large Hi-Energy Obs B (Magn Spect)	6.97	4	Two pair roll out
Large Hi-Energy Obs D (X-ray)	6.97	4	Two pair roll out
Solar Max Satellite	4.46	2	Rigid panel, folded
Gravity and Relativity Satellite	0.84	4	Rigid panel, deployed
Adv Syn Meteorology Sat	4.64	2	Rigid panel, folded
Earth Obs Sat	≈17.65	1	Rigid panel, folded
Synchr Earth Obs Sat	2.79	2	Rigid panel, deployed
Special Purpose EOS	1.80	2	Rigid panel, folded
TIROS O	≈25.82	1	Rigid panel, folded
Environ Monitor Sat	≈10.87	1	Rigid panel, folded
Mars Surf Sample Return	≈6.04	4	Rigid panel, folded
Pioneer Venus Multiprobe	1.11	2	Roll out
Jupiter Orbiter	92.89	2	Roll out
Encke Rendezvous	92.89	2	Roll out
Encke Slow Fly By	92.89	2	Roll out
Intelstat	34.37	2	Roll out

TABLE II (Cont'd)

<u>Payload</u>	<u>Array Size (m²)</u>	<u>Number Required</u>	<u>Deployment Concept</u>
DOMSTAT B	34.37	2	Roll out
Disaster Warning Sat	41.80	2	Roll out
Traffic Mgt Sat	5.57	2	Rigid panel, folded
Foreign Communication Sat	3.16	2	Rigid panel, folded
DOMSAT C	2.14	2	Rigid panel, folded
Lunar Orbiter	3.16	2	Rigid panel, deployed

TABLE III. DEPLOYABLE SUN SHADES

<u>Payload or Mission</u>	<u>Sunshade Diameter and Length (m)</u>
LST	3.66 D x 6.80
Large X-ray Facility	3.66 D x 5.03
Extended X-ray Surv	0.76 D x 1.83
Large Hi-Energy Obs D (X-ray)	1.52 D x 4.11
1.5-m Cryo-Cooled IR Telescope	2.41 D x 1.22
1.0-m Diff Lim UV Optical Telescope	1.65 D x 1.65
3-m Ambient-Temp IR Telescope	3.66 D x 6.80
2.5-m Cryo-Cooled IR Telescope	2.80 D x 1.43
1.5-m Cryo-Cooled IR Telescope	2.41 D x 1.22
1.0-m Diff Lim UV Optical Telescope	1.65 D x 1.65

CATEGORIES OF APPLICATIONS

The survey identified diverse space applications for large-area structures, and they can be separated into the following general categories:

- 1) Large surfaces to provide power
- 2) Large antennas to receive and transmit energy over the radio-frequency band width
- 3) Space platforms to provide area for general utilization

Regarding the first category, specific uses for power or energy include the following:

- . Space manufacturing
- . Power for communications
- . Ion or electric propulsion
- . Subsequent conversion and/or transmission to earth or another spacecraft

Regarding the second category, large radio-frequency antennas are needed for a variety of applications, including:

- . Deep-space communications
- . Multibeam communication satellites
- . Earth and space observations utilizing the antennas as passive radiometers
- . Power transmission in various bands of the spectrum and to either earth or other spacecraft

Finally, in the third category applications exist which would require large structures for docking, assembling, checking out, and resupplying smaller spacecraft or for large work areas (e.g., space processing).

During the survey many of the applications were identified only qualitatively. To make the survey results more meaningful,

analyses were made of the more viable applications to define the approximate sizes of such large-area structures. Those analyses and their results are summarized in the following section.

POWER REQUIREMENTS

Four types of requirements for large amounts of power have been listed in the previous section. The power-station type of requirement would involve by far the largest structures, but, for the reasons stated in the foregoing, will not be examined in detail in this paper. Power for communications, on the other hand, will probably be of relatively small magnitude (tens of kW's at most) and will therefore not be a driving force for developing large power systems. The two remaining types of requirements, power for electrical propulsion and power for space processing, are examined in this section in order to provide an estimate as to the possible future requirements.

Electric Propulsion

In an effort to cause large energy changes with small propellant usage, electric propulsion has been considered for years as a means of achieving high specific impulse. Various types of power sources and thrusters have been envisioned. Current activity centers around the Solar Electric Propulsion Stage (SEPS), which uses an extendible solar-cell array for power generation and mercury ion accelerators for engines (Ref. 2). In its maximum size the SEPS constitutes a propulsion stage which has a power system of 25 kW and seventeen 30-cm-diameter thrusters (Refs. 2 and 3) with a total thrust of 0.93 N. Power and thrust levels can be reduced by shortening the solar panels and by using fewer thrusters. This flexibility of the SEPS configuration allows it to be used for a variety of interplanetary and earth orbital missions. In this paper, attention is concentrated on the earth orbital missions because the survey indicates little need for large power levels for interplanetary types of missions. In earth orbit missions, SEPS is envisioned as being useful, for example, in moving payloads between a Shuttle orbit and geosynchronous orbit. A chemical tug would be used to carry the payload between the Shuttle orbit and the SEPS staging orbit. The SEPS would operate between the SEPS staging orbit and geosynchronous orbit. The SEPS staging orbit would be at an altitude of 16,100 km (10,000 n. mi.), high enough so that SEPS would not be degraded by radiation in the Van Allen belts. Characteristically in this mission SEPS would be able to move 2500 kg of payload to geosynchronous orbit in about 50 days.

The attractiveness of ion propulsion for orbital operations

would be enhanced considerably if the thrust-to-mass ratio could be increased. Much shorter transit times would result, and the variety of payloads which would be aided by ion propulsion would be increased. The manner in which the transit time is affected by the thrust-to-mass ratio is shown in Figure 2. The transit time to geosynchronous orbit is plotted against the thrust-to-mass ratio for two initial orbits: a low earth orbit such as the Shuttle, and a high orbit such as the SEPS staging orbit. The thrust-to-mass ratios considered range upwards by two orders of magnitude from that of SEPS (which is indicated by the shaded area), but are still very small -- small enough so that simplified trajectory analysis can be utilized (see the first section of Appendix A). From this analysis, the transfer time is found to be inversely proportional to the thrust-to-mass ratio. If a thrust-to-mass ratio of 0.01 N/kg could be attained, then the propulsion stage could move the payload from the Shuttle to geosynchronous orbit in five days, or from 10,000 n. mi. to geosynchronous orbit in 30 hours.

These results are based on the assumption that the I_{sp} is large enough so that the propellant use is small. Figure 3 illustrates for the two initial orbits the amount of propellant which would be used as a function of I_{sp} . It can be seen that if I_{sp} is greater than 3000 sec, less than 15 percent of the total mass is expended even for the transfer from the initial Shuttle orbit. Only small errors will result from ignoring a 15-percent mass loss in the trajectory analysis.

In Appendix A, a detailed analysis is made of the performance of ion-propulsion systems. The payload mass fraction is selected as the basis of merit. The payload mass fraction is, of course, dependent on a number of parameters which can be grouped as follows:

- 1) Mission - Orbital change
- Transfer time
- 2) Power system - Output watts per kilogram of power system mass
- 3) Thruster system - Thruster mass per unit exit area
- Density of accelerated ion stream
- Specific impulse

In the analysis a range of power-system effectiveness from 100 to 10,000 W/kg is considered. The lower end of this range

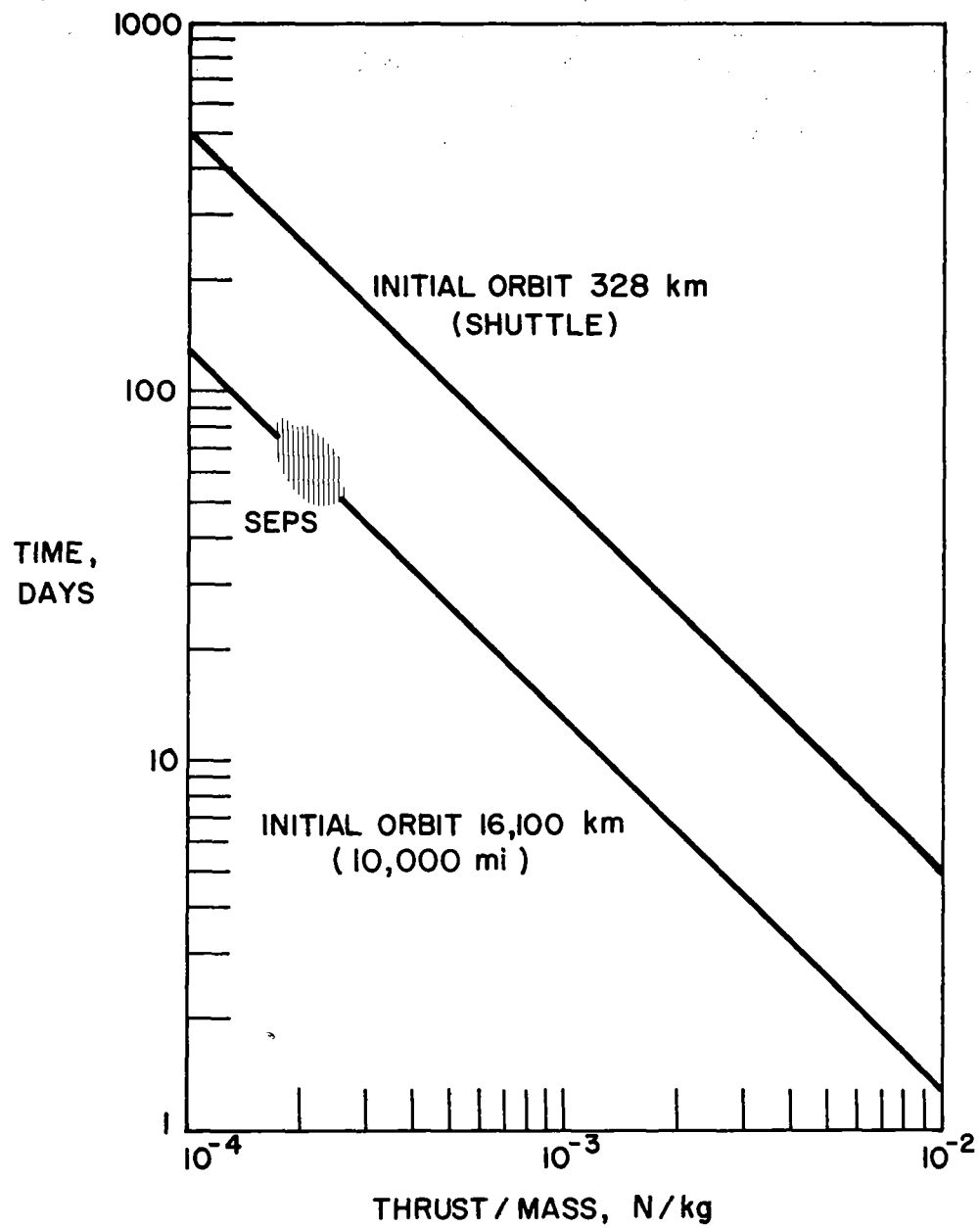


Figure 2. Time to Geosynchronous Orbit, I_{SP} Large

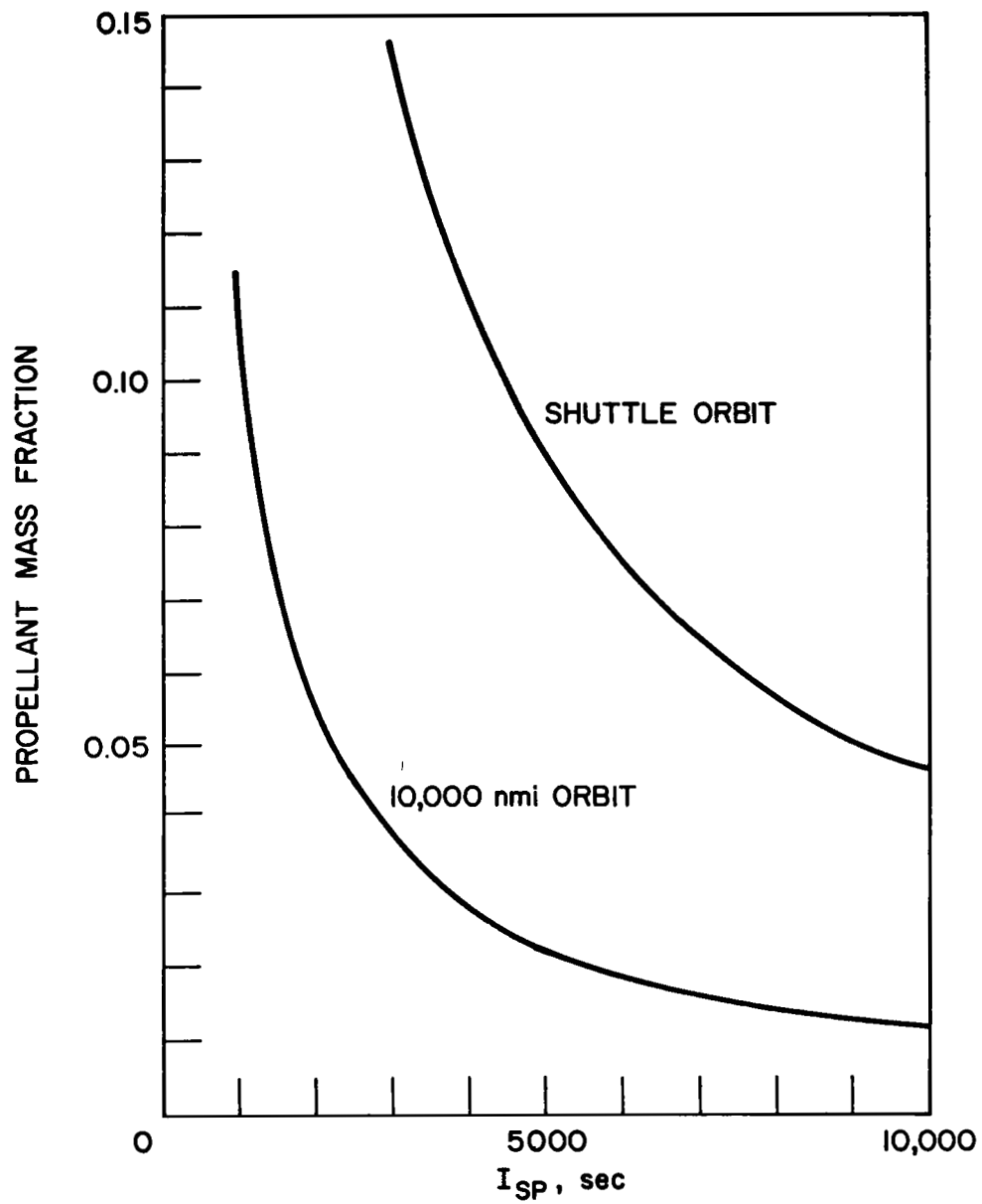


Figure 3. Propellant Use to Geosynchronous Orbit

represents the most advanced technology for which hardware has been developed (Ref. 4).

Thruster values of mass per unit exit area ranging down to 100 kg/m^2 are investigated. This lower value would represent a significant advance over the current technology reported in Reference 4. The types of advance are twofold: one would be the elimination of a heavy dc-ac-dc power conditioner by obtaining high voltage directly from the solar array (Ref. 5); the other would be a threefold improvement in the remaining part of the dry mass.

The analysis is based on mercury as a propellant even though other propellants might become attractive in the future. This choice was made in order that reasonable contact could be maintained with current practice. For mercury propellant an ion-stream density of $9.75 \times 10^{-6}/I_{sp} \text{ kg/m}^3/\text{sec}$ was projected. The value of specific impulse is a matter of choice. Too small a value causes prohibitive thruster weight mass and high propellant usage; too large a value raises the power requirement excessively. So there is a value of I_{sp} which maximizes the payload mass fraction.

Some optimum results are shown in Figure 4. Three missions are considered: five and 50 days from 10,000 n. mi. to geosynchronous orbit and five days from Shuttle orbit to geosynchronous. Note that the five-day high-initial-orbit case has a break in slope due to an arbitrary assumption that no I_{sp} values below 3000 sec would be considered.

Inasmuch as the objective of this analysis is to yield information about the power system, the results in Figure 4 have been replotted in Figure 5 in terms of the amount of power required. Here the total power required to propel a 10,000-kg payload is plotted against the power-system effectiveness for the three missions.

Three interesting possible propulsion-system combinations come from this plot. For the high-initial-orbit, 50-day mission, a power system of around 60 kW with an effectiveness of 100 W/kg would be quite attractive. The vehicle, including its power system, could be called a "large SEPS". For a five-day mission from the high initial orbit, a power system of 800 kW with an effectiveness of 400 W/kg would be an attractive possibility. This propulsion stage could be termed an "advanced rapid SEPS". Finally, a power system of about 10 MW with an advanced effectiveness of 2000 W/kg could move 10,000 kg of payload from the Shuttle

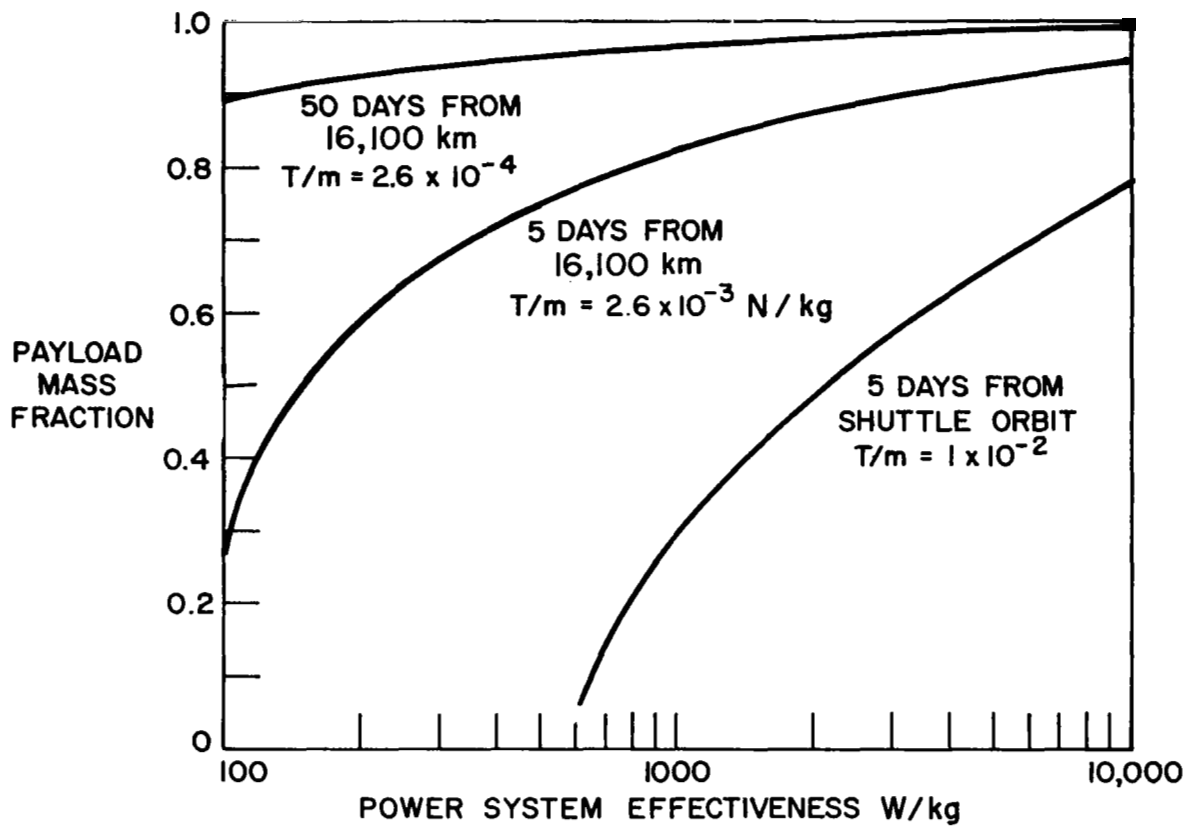


Figure 4. Payload Mass Fractions
Versus Power-System Effectiveness
For Various Electric-Propulsion Systems
To Reach Synchronous Orbit

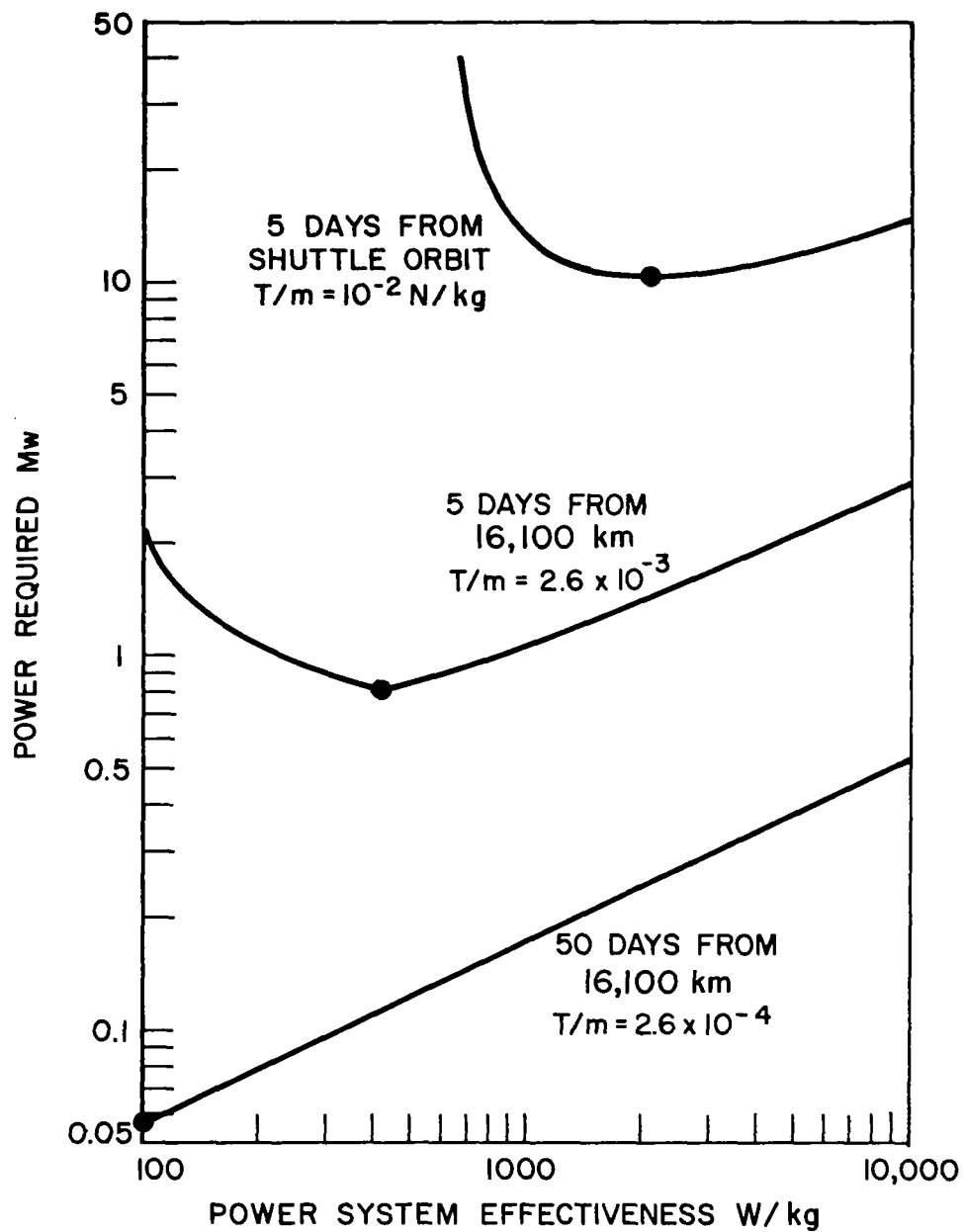


Figure 5. Advanced Orbital Transporters,
10,000-kg Payload to Geosynchronous Orbit,
Advanced Thrusters

to geosynchronous orbit in five days. This could be termed a "rapid electric tug". The mass breakdown and other characteristics of these three vehicles are shown in Table IV.

These three propulsion systems provide a basis for possible power system requirements in the future. These requirements will be summarized, along with those arising from space processing, in a later section.

Space Processing

One of the most exciting possibilities for economic usefulness of space flight is that of applying the unique characteristics of space to produce products useful here on earth. The possibility exists that some currently useful products can be made with less expense or higher quality in space than here on earth, but the most interesting possibility is that totally new products will be made feasible and that these new products will have a profound effect on our lives. A comparable historical example is that of solid-state electronics which was undreamed of before the invention of the transistor in the late 1940s but which now pervades almost every facet of our lives.

The principal unique characteristic that space has to offer from a processing standpoint is ultra-low environmental stress. It is this heretofore unavailable characteristic that can be expected to lead to new products. Examination of technical history indicates that a new capability is usually followed by a related advance. History can be expected to repeat itself in the case of space processing.

NASA has been performing and supporting studies on space processing for some time. The most detailed of these studies with application to future economic projections is that being conducted by GE (Ref. 6). This study has delved into processing details such as batch sizes, cycle times, temperatures, and power levels for dozens of possible types of products. For the purpose of the present survey the important outcome of these and other studies is the impressive array of possible products in significant volume and the need for power to perform the processing. When space processing becomes a usual occurrence, power stations will be required to supply the necessary power in space for processing.

The size of the space power supply is certainly interesting to consider. The GE studies indicate that at least 50 kW will be

TABLE IV. ADVANCED ORBITAL TRANSPORTERS FOR 10,000-KG PAYLOADS

	Large SEPS, 10,000 n. mi. to geosynchronous orbit in 50 days	Advanced Rapid SEPS, 10,000 n. mi. to geosynchronous orbit in five days	Rapid Electric Tug, 200 n. mi. to geosynchronous orbit in five days
Power	56.6 kW	782 kW	10.5 MW
I_{SP}	3000 sec	3000 sec	6300 sec
M_{power}	566 kg	1954 kg	5260 kg
$M_{thruster}$	110 kg	1383 kg	3720 kg
$M_{propellant}$	456 kg	571 kg	1540 kg
$M_{payload}$	10,000 kg	10,000 kg	10,000 kg
M_{total}	11,132 kg	13,908 kg	20,520 kg

required to process a reasonable batch of material. Some calculations based on being able to produce significant quantities of material in a reasonable time indicate the need for multiple parallel batches with an order-of-magnitude increase in average power requirement.

An independent estimate of the required power levels can be obtained from Figure 6, which shows the amount of power radiated from molten spherical masses of various materials. It can be seen that in order to hold a kilogram of tungsten in a molten condition, almost 100 kW of power is required, assuming that other process losses are counterbalanced by the back radiation of surrounding walls of the processing facility. If this power level is required for 15 minutes to complete the process, then 100 kW could produce only about 700 kg of tungsten per week. If it is desired to process "Shuttle loads" in a reasonable length of time, a megawatt or more would be required.

The foregoing straightforward calculation yields results in agreement with those of other studies and indicates the likely need for megawatt-size space power supplies to support processing of products in space.

Structural Concept Implications

The power requirements generated in the preceding sections are summarized conveniently in Figure 7. In preparing this figure, the assumption has been made that the power source is the sun. The space power system therefore collects solar energy and converts it to electrical energy. In the figure, the power level is expressed as a "size" which is the square root of the required area of the solar collector. This size is plotted as the ordinate in the figure; the abscissa is the power-system effectiveness as measured in watts of output power per kilogram of power system mass. In converting the power levels to size, a value of 100 W/m^2 of solar collector is assumed for the lower values of effectiveness. This number is characteristic of photovoltaic conversion systems (solar cells). For the higher values of effectiveness ($\approx 1000 \text{ W/kg}$) a conversion value of 200 W/m^2 is used in recognition that such high effectiveness can probably be attained only through more efficient power-conversion devices.

Trends in power requirements resulting from the preceding analyses are shown by the shaded areas in the figure. For electric propulsion, the points within the shaded area indicate the three

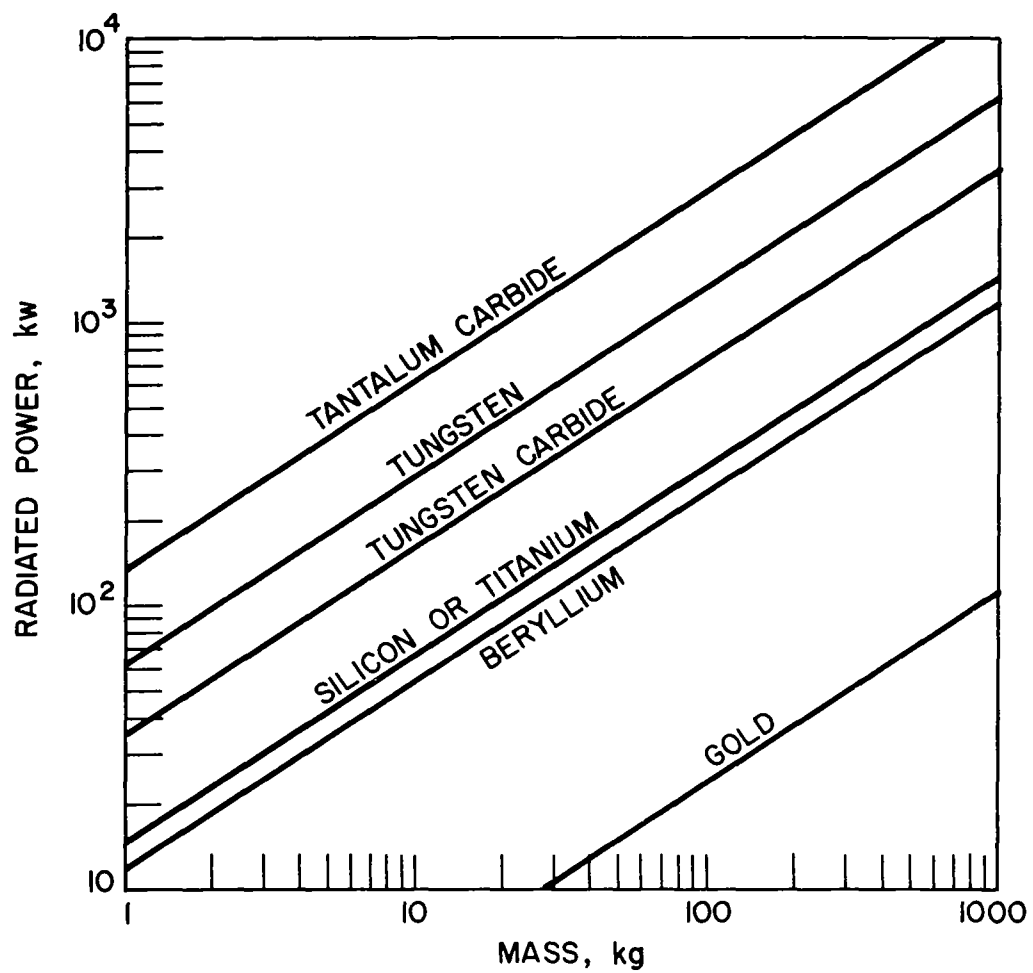


Figure 6. Power Required for Melting

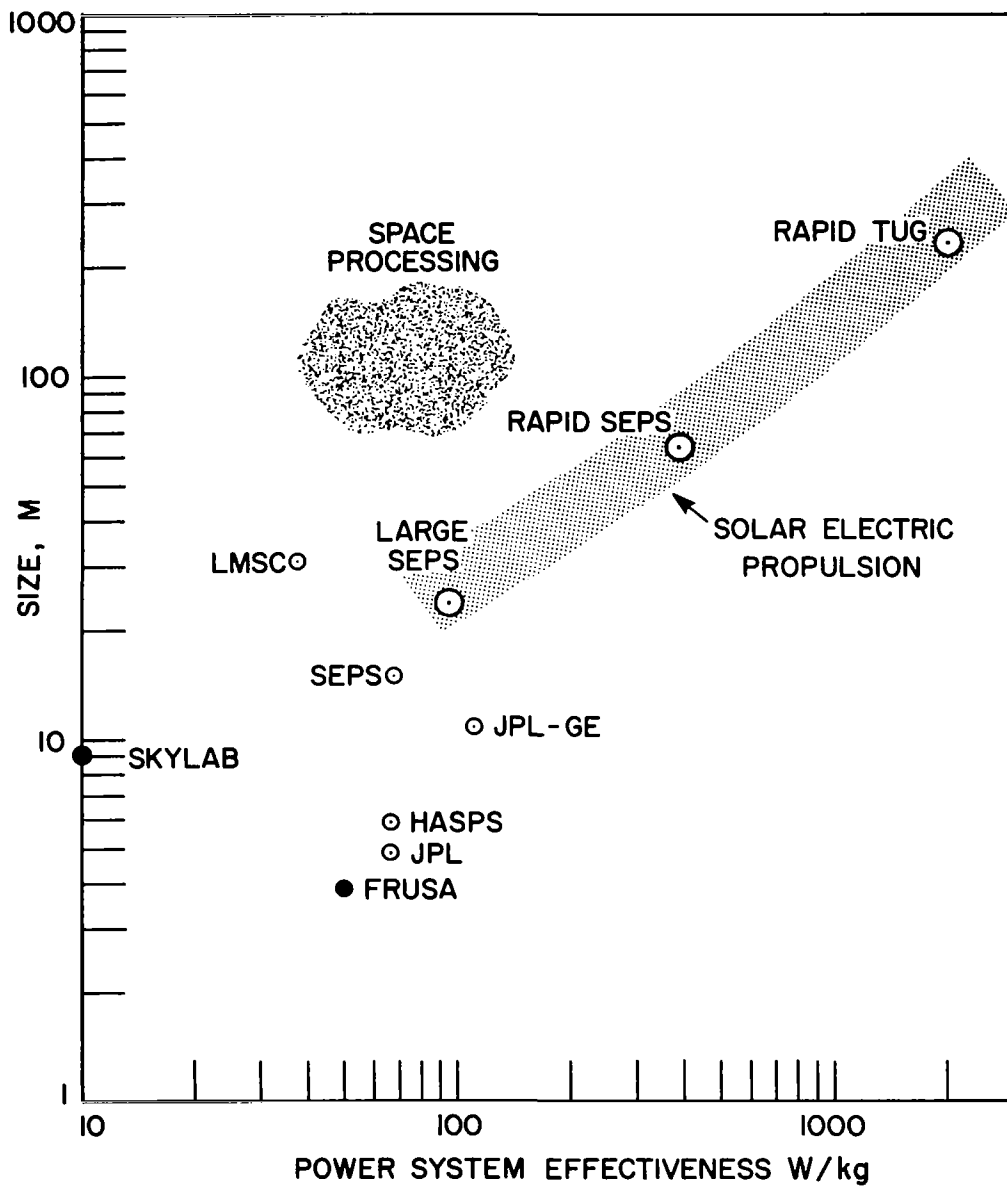


Figure 7. Solar Power Systems

propulsion vehicles identified in the preceding section. There is a trend toward higher effectiveness and larger sizes as propulsion performance increases. The demands of advanced electric propulsion on power-system effectiveness are very severe, as can be seen by comparing them with the current technology indicated by the various labeled symbols on the figure. Current technology is bounded by 100 W/kg in effectiveness and 10 to 30 m in size. An order-of-magnitude increase in both size and effectiveness is needed to allow large advances in electric propulsion beyond SEPS. On the other hand, the space-processing power requirement has no strong need for high effectiveness. Therefore the shaded area shown for space processing includes those effectiveness values which are attainable by only moderate advances in the current technology and is limited to a size which can be packaged as a single Shuttle payload.

ANTENNA REQUIREMENTS

Large antennas are, of course, large-area structures. The most prevalent type of large antenna is characterized as a surface (reflector, lens, phased array) from which radiation emanates in a controlled fashion. In order that the antenna perform its desired use, the frequency, polarization, intensity, and wave-front phase of the radiation must be carefully controlled in a predetermined manner. Intensity, polarization, and frequency usually offer little difficulty from a structural point of view; however, the wave-front phase of the radiated signal is dependent on the geometric position of the radiating surface elements and is therefore highly influenced by structural accuracy. Large antennas impose very large demands on structural geometric stability. The allowable error in phase ranges from $1/25$ wavelength when the energy in the side lobes is of concern, to $1/4$ wavelength when gross deterioration occurs in the main beam gain.

The size of the antenna surface determines the basic angular dimension of the main radiated beam. The half-power beam width of the main beam is approximately the wavelength divided by the size of the aperture. Since most of the radiated power is contained in the main beam, the gain is approximately inversely proportional to the square of the beam width, or is proportional to the antenna area divided by the square of the wavelength. The shaping of the main beam is dependent on the distribution of intensity across the antenna surface. Beam patterns are ordinarily twice as wide as the half-power beam width, but by tapering the intensity near the edges of the aperture, the beam pattern can be made more uniform in the center, with a much more rapid drop near the edges of the beam.

Various bands of frequencies are used for various types of applications. A system of letter-type designations for the various frequency bands has evolved over the years as a convenience. Following is a list of the designations and corresponding frequencies and wavelengths.

<u>Name of Band</u>	<u>Frequency (GHz)</u>	<u>Wavelength (cm)</u>
vhf	0.10	300
uhf	0.7	43
L	1.5	20
S	3.0	10

<u>Name of Band</u>	<u>Frequency (GHz)</u>	<u>Wavelength (cm)</u>
C	6.0	5
X	11.0	2.7
K _U	17.0	1.8

The vhf and uhf bands are used for broadcast communications work; L-band is used for navigation and telemetry; S-band is used by NASA for its deep-space communications; C-band is used worldwide for communications satellite purposes; and X- and K-bands are beginning to be used for a variety of purposes because of the overcrowding of the lower frequencies of the electromagnetic spectrum.

Generally speaking, as frequency is increased, the electronic components become more difficult to build. On the other hand, the antennas become smaller and lighter, although they need to be of higher accuracy and so are also more difficult to build.

There are three main types of surface-type antennas. In the reflector type the energy is radiated from a relatively small source and is reflected by an essentially paraboloidal surface to form a collimated beam. A reflector has the advantages of a passive large surface and the ability to handle a range of wavelengths (broad-band). Its major disadvantage is that any structural distortion affects the phase of the wave front directly. Indeed, the phase error is twice the structural distortion.

In a lens antenna the energy is also emanated from a small source but is collimated by a lens made of dielectric material. Lenses are passive but tend to be quite thick and heavy. They can be zoned (as in a Fresnel lens) to reduce their volume and weight, but then they become narrow-band. On the other hand, relatively large structural distortions can occur in a lens before the wave-front phase becomes seriously degraded. Phase errors are characteristically an order of magnitude less than the structural distortions.

The phased array is a distribution of active, directly radiating elements over the large surface of the antenna. Phased arrays can be built to be broad-band but are complex and therefore expensive because of their widespread use of active electronics. They can, however, be made adjustable in phase so that techniques can be devised to correct for any structural distortion by purely electronic means.

If lightweight low-cost antennas are desired, reflectors are

to be preferred. As demands for accuracy increase, there are strong arguments in favor of lenses for low-cost applications, and in favor of phased arrays for extremely accurate and flexible operations. The requirements on size and accuracy arise from the mission application of the antenna. In the following sections, two uses are examined, the first being multibeam communication satellites and the second being microwave radiometry.

Multibeam Communication Satellites

The use of geosynchronous-orbit communication satellites has proven to be an outstandingly cost-effective application of space flight. A current satellite costs \$25 million including launch costs, and generates \$20 million per year in gross income for a life of five years. The availability and quality of satellite communication have already had a marked effect on our lives.

Projections into the future indicate an enormous increase in communications traffic. In a NASA-supported study of geosynchronous platforms performed by Rockwell International, the predicted worldwide requirement is an average of 600,000 channels of voice communication by the year 2000. This means that an average of 4×10^{10} bits/sec will be required. Current satellites can handle 4×10^8 bits/sec, so 100 would be needed. Even for advanced satellites that could handle 10^9 bits/sec, 40 would be needed and space in geosynchronous orbit could become crowded.

The problem could be even more severe. Suppose that all the peoples in the world want to interact with each other (telephone, television, travel, etc.) as the North Atlantic peoples will want to. Extending even the current travel frequency of people in the advanced nations to the world population yields a prohibitively large demand on energy for transportation. The only reasonable alternative seems to be to provide excellent communication links so that the interactions can occur with low energy expenditure.

In order to be accepted as an alternative to physical travel, the communication links would have to be as available as the telephone is today but with much wider band width. In short, party-to-party videophone is required. With a TV band width the requirement for 600,000 channels grows to the order of 4×10^{12} bits/sec. The result is that the satellites would have to be extremely advanced, with capability of switching on the satellite

and with antennas which can emit very tight beams to link with individual users (or, at most, neighborhoods) on the ground.

For the purposes of the present study, interest is concentrated on the beam size. If each beam is 10 MHz in band width and a total band width of 1 GHz is available, then only 100 customers could be served simultaneously by one beam. In an urban situation, a beam diameter of 10 km appears to be an appropriate order of magnitude. Beam shaping would be required to reduce cross talk.

In order to meet the requirement of a shaped 10-km beam from geosynchronous orbit, an antenna 100 m in diameter is required at 20 GHz. The wave-front phase will need to be accurate to 1 mm. This requirement is summarized later in this report.

Microwave Radiometry

Space flight creates the capability of observing man's surroundings from a global viewpoint. Contrary to some early predictions, observation from orbit is often quicker, cheaper, and more informative than ground or aircraft surveys and measurements. One of the reasons that so much useful data are yielded by a single observation satellite is the multispectral character of the observation. Almost all observations currently being made are in the visible and infrared regions. Extending the spectrum into the microwave region would greatly add to the capability of remote observation. The economic advantages of earth observation from space will inexorably supply the necessary pressures for the advances in technology required for that extension.

Observation of the earth from space can be either active or passive. Observation in the microwave wavelengths has been limited almost entirely to active radar observations which require large amounts of power in the sensor. Passive observation, which requires no radiated power, has been relatively unexplored, presumably because of the very small amounts of energy involved in the natural radiation from bodies in the radio frequency regime and the large antenna sizes required for reasonable resolution.

Passive observation at radio frequencies is termed microwave radiometry. The procedure is to measure the amount of power that is received by the antenna within a particular frequency band. The power measured is usually expressed in terms of an apparent temperature in accordance with the formula:

$$\frac{P}{B} = K(\epsilon T)$$

where P = power received by the antenna in band width B

K = Boltzmann's constant

T = absolute temperature

and ϵ = emissivity

The product ϵT is the apparent temperature. The temperature thus measured would be the actual temperature of the object observed if the radiation from the object were "blackbody" (emissivity of 1). In fact, the radiation is not blackbody and indeed the emissivity varies a great deal more than the temperature itself. Microwave radiometry, therefore, is essentially a technique of measuring the emissivity of the observed body at various frequencies.

There are a number of interesting phenomena which can be observed remotely by determining the emissivity at radio frequencies. Table V contains a summary of some of these phenomena together with information about the frequency at which the phenomena can be examined and the resolution on the earth's surface that is considered to be desirable (see Appendix B for a more detailed discussion of observables). The frequencies of interest range from 1 to 30 GHz. At the lower frequencies the amount of moisture in the soil at depths of several meters can be determined since the electrical conductivity and hence the skin depth and emissivity at those frequencies are influenced by the amount of moisture. At the higher end of the frequency range, useful information can be obtained such as the location of areas of precipitation. Since the surface of the earth has a rather low emissivity, the presence of a thick layer of water droplets or snow particles increases the apparent emissivity considerably. Precipitation, therefore, appears to be considerably warmer than the earth or oceans even though the particulate matter actually may be of lower temperature.

The technology of measuring the power in various radio-frequency bands has advanced to the point where apparent temperature differences of a degree Kelvin are measurable. The success of microwave radiometry, therefore, hinges on the ability of the sensing antenna to provide a narrow enough beam so that the desired resolution is obtained. Not only must the beam be narrow, but also the side lobes must be very low so that the power received by the antenna comes only from the main lobe. For observation, even from low earth orbit, the antenna size for investigation of soil moisture can be as large as 1 km in diameter. The identification

TABLE V. MICROWAVE RADIOMETRY

<u>Application</u>	<u>Frequency (GHz)</u>	<u>Resolution (km)</u>
Soil Moisture	1 - 10	0.2
Dynamic Ocean Surface	≈ 10	100
Ocean Surface Temperature	5	5 - 150
Salinity	≈ 1	< 5
Sea Ice	3 - 30	0.2
Storm Cells	≈ 30	2

of precipitation areas from geosynchronous orbit would require a 300-m-diameter antenna with a surface accuracy of better than 1 mm. Microwave radiometry appears to be a strong requirement for large space structures and one which demands a considerable increase in the state of the art.

Antenna Summary - Structures Implications

Figure 8 is a graph showing the required size and frequency ranges for the previously discussed space antenna reflectors. Also in this figure, the future requirements are compared with existing reflector properties, clearly showing that future reflectors must be larger, more accurately shaped, and more dimensionally stable. These improvements are needed to provide narrower beam widths, greater gain, and the increased band widths available at higher frequencies. Space reflectors to concentrate solar power and to transmit and receive power have these same requirement trends, although they are not reviewed in this paper.

These size and shape requirements are not necessarily restricted to reflectors; lenses and phased arrays, and perhaps as-yet-undefined concepts, might also be developed to meet these requirements. However, only the development of the indicated reflectors is clearly within the area of expertise claimed by the structures community.

Such reflectors must meet other structural requirements:

- 1) They must be lightweight so that transporting them to space is economically feasible;
- 2) They must be transportable by available vehicles such as the Space Shuttle; therefore, they must in some manner be reducible from full size (i.e., they must be modular, foldable, able to be fabricated in space from raw materials, or other);
- 3) Once transported to space, they must be deployable by such means as automatic erection, use of remote manipulators, or astronaut EVA;
- 4) Once deployed, they must be dimensionally stable and/or adjustable.

Other requirements exist (for instance, for pointing and feed capabilities), but they are not so clearly within the purview of structures technology.

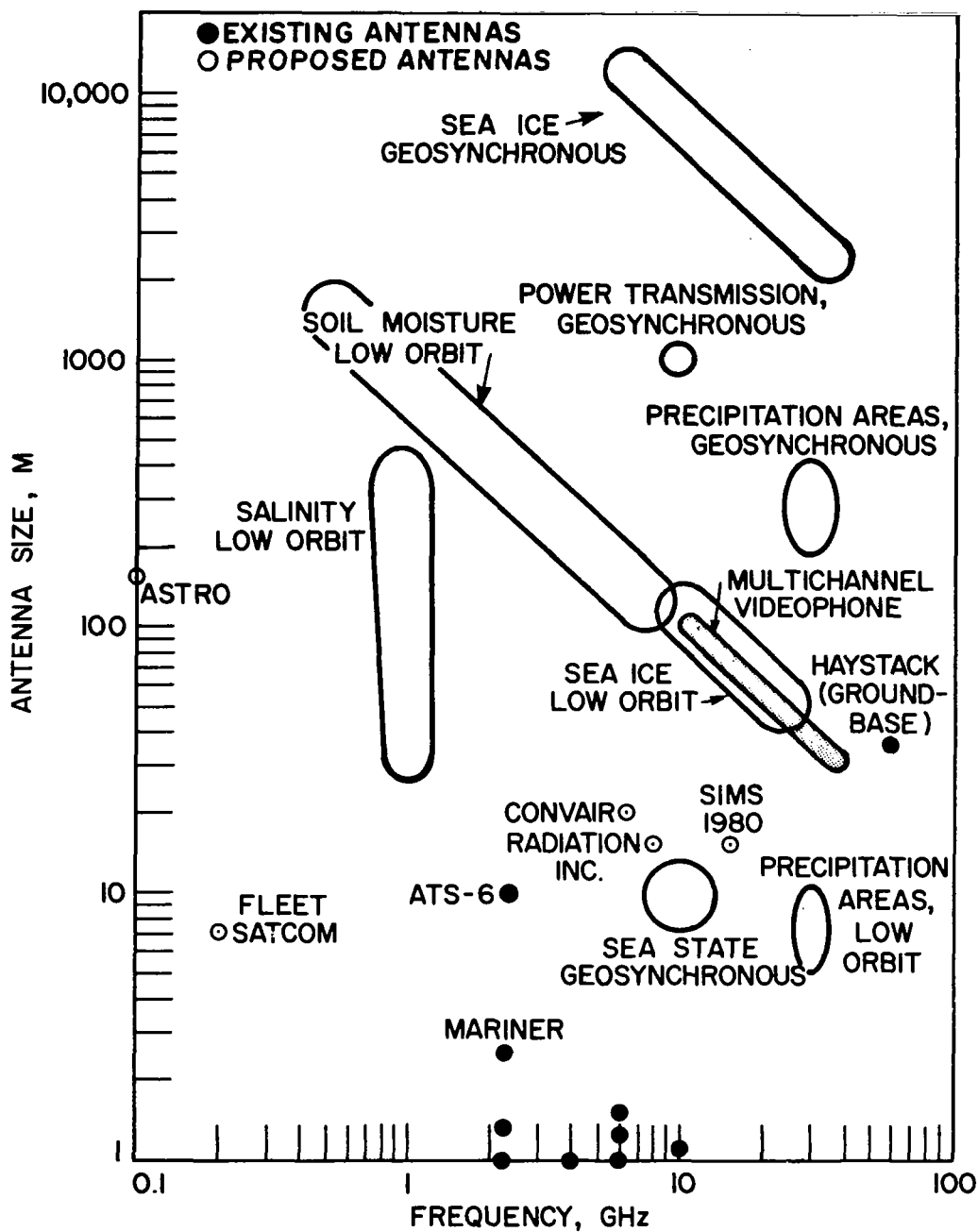


Figure 8. Size Versus Frequency Requirements For Future Antenna Applications

No existing structural approach appears to be directly adaptable to the increased reflector requirements. Lockheed, Missiles and Space Company, Inc., has developed a 30-ft-diameter furlable mesh reflector (Ref. 7) which is now orbiting on the ATS-6 satellite. Harris Corporation is developing for NASA, Langley, a mesh reflector similar to the Lockheed system but with potentiality for more accurate shaping. However, both the Lockheed and Harris reflectors are supported by deployable radial ribs. If the diameters of these concepts were scaled to the 100-m range, these ribs would become impractically numerous. General Dynamics (Ref. 8) has developed another mesh-type reflector that is deployed and supported by a three-dimensional trusswork. However, it too appears to have an impractically large number of parts to be used as a 100-m, accurately shaped reflector. Therefore, new structural concepts will be required for reflectors.

As discussed earlier, lenses show certain advantages over reflectors. However, the practicality of using large lenses depends on development of a lightweight dielectric material and on development of means to deploy such lenses, and no work along these lines has been performed.

BASELINE APPLICATIONS

To limit the scope of future conceptual studies of large-area space structures, the following baseline applications were selected, which typify the various kinds of future applications revealed by the survey.

<u>Application</u>	<u>Requirements</u>
1. Solar-power collector	Power level = 1 MW
1.1 With solar cells	Efficiency = 100 W/kg
1.2 With solar concentrator	Efficiency > 400 W/kg
2. Large antenna reflector	Diameter = 100 m
2.1 For radiometry	Operating limit \approx 10 GHz
2.2 For geosynchronous communications	Operating limit \approx 20 GHz
3. Space platform	Planar area = 100 m x 50 m
	Alignment accuracy < 10 arc sec

The reasons for selecting these particular applications as baselines are discussed below.

Solar-Power Collector

Both the "Electric Propulsion" and "Space Processing" sections of this report show that power levels of the order of one megawatt will be required for solar-power collectors. Figure 7 shows that if a collector provides power at efficiencies greater than 100 W/kg, say 400 W/kg, that power will be suitable for even a "rapid SEPS" type of ion-propelled spacecraft. Solar cells can be assembled in arrays large enough (10^4 sq m of conventional cells) to generate one megawatt. Further, the efficiency of an advanced solar-cell array could be at least 100 W/kg, as projected by the JPL-GE study. Accordingly, a baseline one-megawatt solar array is selected to meet the needs of an ion tug as well as some of the needs of the space-processing activities.

As discussed in the "Power Requirements" section, conventional solar-cell technology does not show as much potentiality for

efficiently providing solar power as do other collecting and converting methods, which generally require solar-power concentration. Thus, the second baseline application for solar-power collectors is a concentrator. Note that a solar concentrator need not be as accurate as the diffraction-limited antenna reflectors specified as baseline antenna reflectors.

Large Antenna Reflector

Although two different baseline antenna reflectors are listed above, both can be met by the same advancement in reflector technology. That technology level is seen in Figure 8 to be applicable both to several of the radiometry applications and to geosynchronous multichannel communications which require spatial separation of beams. Moreover, the specifications for the baseline reflectors call for an advancement in the state of the art of about an order of magnitude. The state of the art as understood here and implied by Figure 8 is essentially for reflectors of diffraction-limited performance. In the case of the baseline communications antenna, the rms surface imperfection should be about one-twentyfifth of the carrier wavelength. But in the radiometer application, imperfections should be less than about one-fiftieth of the nominal wavelength being sensed.

Space Platform

The space-platform baseline application is selected as a foundation for fabrication, assembly, or alignment of spacecraft or components. The 10-arc-second accuracy of the 100-m by 50-m planar surface is feasible inasmuch as such accuracies can be measured, for instance, by existing vector magnetometers.

This large, accurate planar surface is expected to be especially useful for aligning components of assembled-in-space spacecraft which are either too heavy or too large to be assembled on earth. It is also expected that such a platform might be very useful in the space assembly and alignment of the large apertures referred to earlier.

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APPENDIX A - ION PROPULSION

Simplified Trajectory Analysis

For a small mass m in motion about a large gravitating body and acted upon by radial force F_r and tangential force F_θ , dynamic equilibrium is given by

$$\ddot{r} - r\dot{\theta}^2 = -\frac{\mu}{r^2} + \frac{F_r}{m} \quad (A-1)$$

and

$$r\ddot{\theta} + 2\dot{r}\dot{\theta} = \frac{F_\theta}{m} \quad (A-2)$$

In these equations,

r = radial distance from center of gravitation to mass m

θ = angular position of mass

μ = gravitational constant, $3.99 \times 10^{14} \text{ m}^3/\text{sec}^2$

When $F_r = 0$ and the mass is in a circular orbit ($\ddot{r} = 0$), equation A-1 yields

$$\dot{\theta} = \sqrt{\frac{\mu}{r^3}} \quad (A-3)$$

Then

$$\ddot{\theta} = -\frac{3}{2} \dot{r} \sqrt{\frac{\mu}{r^5}} \quad (A-4)$$

When these relationships for $\dot{\theta}$ and $\ddot{\theta}$ are substituted in equation A-2 it becomes

$$\frac{F_\theta}{m} = \frac{\dot{r}}{2} \sqrt{\frac{\mu}{r^3}} \quad (A-5)$$

It is now assumed that F_θ is constant and due to steady ejection of a portion of m ;

$$F_\theta = \dot{m} I_{sp} g \quad (A-6)$$

where I_{SP} = specific impulse of the ejected mass.

The combination of equations A-5 and A-6 then gives

$$\frac{\dot{m}}{m} = \frac{\dot{r}}{2I_{SP}g} \sqrt{\frac{\mu}{r^3}} \quad (A-7)$$

where $m = m_0 - \dot{m}t$.

By integrating equation A-7 from time $t = 0$ when $r = r_0$ and $m = m_0$, to time $t = t_1$ when $r = r_1$ and $m = m_0 - m_f$, we obtain

$$\frac{m_f}{m_0} = 1 - e^{-\frac{1}{I_{SP}g} \left(\sqrt{\frac{\mu}{r_0}} - \sqrt{\frac{\mu}{r_1}} \right)} \quad (A-8)$$

where the propellant mass m_f is

$$m_f = \dot{m}t_1 \quad (A-9)$$

To indicate the validity of this simplified analysis, consider radial accelerations due to tangential thrusting as compared to radial accelerations due to gravity. By combining equation A-5 with its time derivative, the radial acceleration due to thrusting is found as

$$\ddot{r} = 6 \left(\frac{T}{m} \right)^2 \frac{r^2}{\mu} \quad (A-10)$$

where $T = F_\theta$ = thrust.

Since the gravitational acceleration is μ/r^2 , the ratio λ of the two accelerations is

$$\lambda = 6 \left(\frac{T}{m} \frac{r^2}{\mu} \right)^2 \quad (A-11)$$

For the maximum value of T_θ/m considered here, that is 10^{-2} N/kg, and for the synchronous radius $r = 4.219 \times 10^6$ m,

$$\lambda = 0.012$$

Therefore, the effect of setting \ddot{r} equal to zero in equation A-1 is negligible for the present calculations.

For large values of specific impulse I_{SP} , the propellant mass fraction ejected during transit to synchronous orbit is small, as shown in Figure 3 of the main text. The time to achieve synchronous orbit is then obtained from equation A-6 as

$$t_1 = \frac{m_f}{m_o} \frac{m_o}{T} I_{SP} g \quad (A-12)$$

where values of m_f/m_o are calculated from equation A-8, and values of T/m_o are in accordance with the assumed state of the art.

Payload Mass Fraction Analysis

The general data on ion engines used here were obtained principally from References A-1 through A-8, and the specific data on future ion-propulsion systems that do not use dc-ac-dc power converters were obtained from Reference A-8.

The total initial mass m_o of an ion-propelled system is assumed to be

$$m_o = m_{PL} + m_P + m_f + m_T + m_{PC} \quad (A-13)$$

where m_{PL} = payload mass

m_P = power-collector mass

m_f = mass of propellant

m_T = thruster mass

and m_{PC} = power-conditioner mass

The mass of the power-collector subsystem is assumed to be directly related to power input;

$$m_P = \frac{\text{power}}{K_1}$$

$$\text{or} \quad = \frac{TI_{SP} g}{2\eta K_1} \quad (A-14)$$

In this equation η is the power conversion efficiency for

mercury-ion engines and is estimated from the cited references to be about 0.65 . The term K_1 expresses the efficiency with which power is collected and converted to electricity. The values considered for K_1 range from 20 to 2000 W/kg . This range includes values for present and advanced technology for solar-power conversion. Also, the K_1 values take into consideration the masses of the solar collector, electrical generator, switching device, and distribution network.

The propellant mass is taken from equation A-12 as

$$m_f = t_1 \frac{T}{m_0} \frac{m_o}{I_{SP}g} \quad (A-15)$$

The mass of the ion engine (thruster) is expressed as the product of the engine exit area A and factor K_2 . This product expresses the mass of the engine per unit exit area as

$$m_T = K_2 A \quad (A-16)$$

Thrust is related to exit area by the equation

$$T = \rho A (I_{SP}g)^2 \quad (A-17)$$

where ρ = propellant density.

Under the conditions that the current density is limited to 30 A/m² (in order to provide for long grid life), and that only mercury vapor propellant is considered, then, by utilizing the Child-Langmuir law (Ref. A-3), density is expressed in the following form:

$$\rho = \frac{K_3}{I_{SP}} \quad (A-18)$$

Therefore, m_T is obtained in the form:

$$m_T = \frac{K_2 T}{K_3 I_{SP} g^2} \quad (A-19)$$

Values of K_2 from 100 to 300 kg/m² are considered as representative; 300 kg/m² is approximately the efficiency of the NASA SEPS vehicle which has an exit area of 0.0707 m² . As used here, K_2 includes the combined masses of the engine chamber, gimbals, and

engine support structure. A value of

$$K_3 = 9.75 \times 10^{-6} \text{ kg-sec/m}^2$$

is appropriate for mercury engines (to which the present study restricts itself).

In Figures 4 and 5 of the main text it is assumed that no power conditioner is used. However, for possible future reference, power conditioners are discussed below. The mass of a dc-ac-dc power conditioner is closely approximated by

$$m_{PC} = 0.59 P^{0.48} \text{ (kg)} \quad (\text{A-20})$$

where P = power (W).

Technical discussions with personnel at NASA Lewis Research Center indicate that power conditioners for two recent experimental ion engines (with 8- and 30-cm throat diameters) are at least eight times as massive as the engines themselves. This is a substantial increase over the ratios indicated in Reference A-1, and is the penalty associated with larger temperature variations in the 8- and 30-cm engines.

Since the ion engine is basically a dc device, and since present-day solar collectors generate dc, deleting the dc-ac-dc converter is feasible. In fact, Reference A-9 describes in detail the advantages of series-coupled cells for achieving a high-voltage solar-cell array (HVSA). Concepts for HVSA include the use of multiple-junction edge-illuminated solar cells (M-J cells) and short circuit diodes. Besides their potential for eliminating the need for power converters, HVSA offer potential for improving power generation efficiency and reliability.

By assuming that $m_{PC} = 0$, and by using the foregoing expressions for the various masses of the ion-propulsion system, the payload mass fraction is obtained as

$$\frac{m_{PL}}{m_0} = 1 - \frac{T}{m_0} \left[\frac{I_{SP} g}{2 \eta K_1} + \frac{t_1 + \frac{K_2}{K_3 g}}{I_{SP} g} \right] \quad (\text{A-21})$$

Figure A-1 is a graph of I_{SP} and K_1 values which maximize m_{PL}/m_0 for 5- and 50-day transit times and for $K_2 = 100 \text{ kg/m}^2$. Values of η and K_3 are as indicated previously. These optimum I_{SP} values are used in preparing Figures 4 and 5 in the main text.

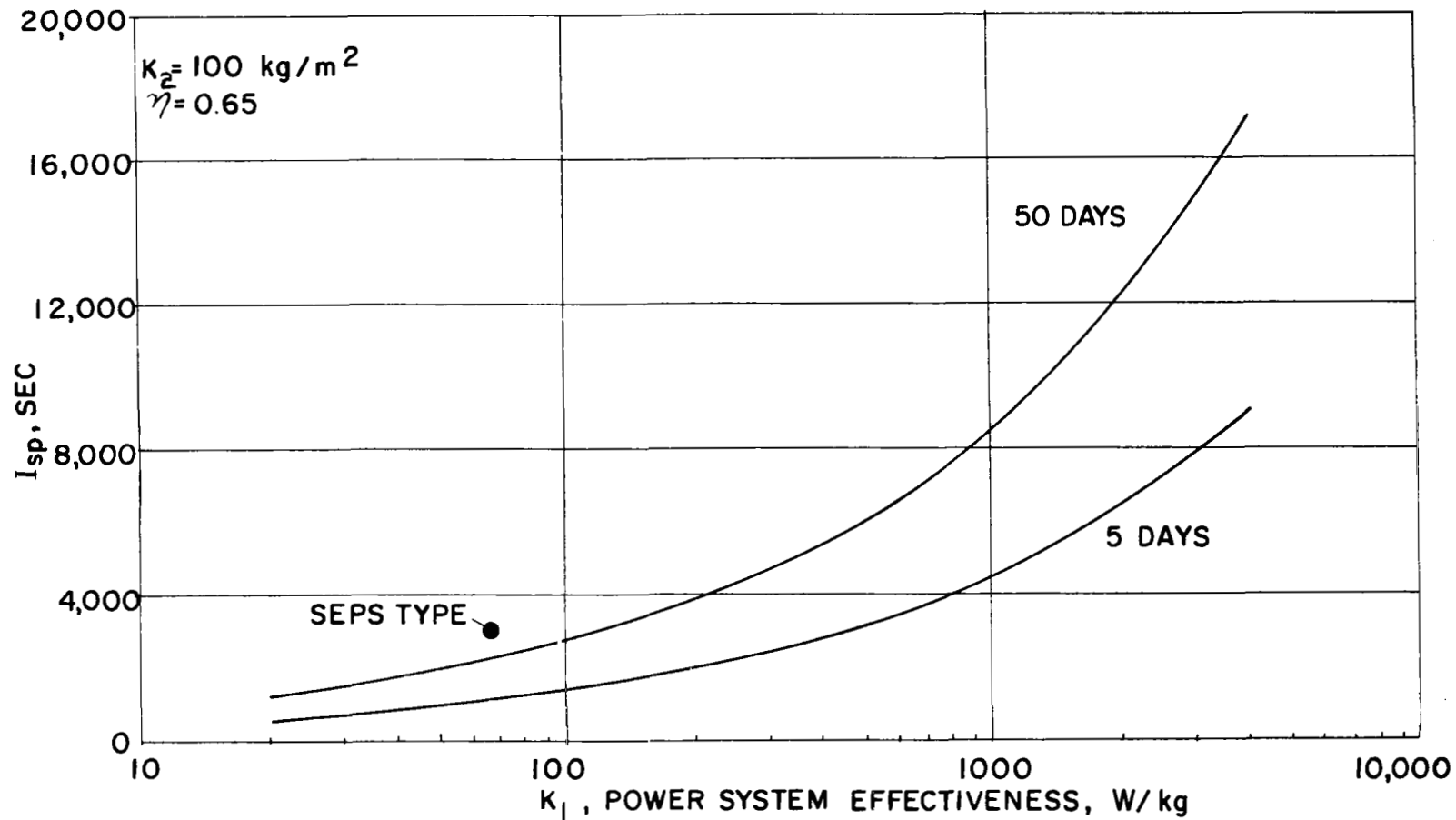


Figure A-1. Specific Impulse and K_1 Values
To Maximize Payload Mass Fraction m_{PL}/m_o

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APPENDIX B - MICROWAVE RADIOMETRY

For some time orbiting infrared radiometers have been used to observe passively earth resources and properties. However, present technology indicates that microwave radiometers can clearly provide much more data. This is because it is now known that many observables emit energies in distinctly different band widths in the microwave range, and that the earth atmosphere is relatively transparent to those band widths.

Figure B-1 shows attenuation of radiation through the earth atmosphere as it varies with radiation frequency for both humid and dry conditions. Frequencies in the range of 1 to 20 GHz (30- to 1.5-cm wavelengths) clearly offer observation potentiality because of low attenuation.

Microwave earth observations from orbital altitudes will require radiometers (antennas) that use large and accurately shaped parabolic reflectors. The reflectors must be large enough to locate sources accurately and to provide sufficient gain to detect emissions. However, size alone is not sufficient; the reflectors must also be accurately shaped.

The microwave power density radiated from typically observed earth objects is small. Therefore, the effectiveness of a microwave radiometer depends upon its gain, its capability to accept power over a wide band width, and its capability to integrate weak signals timewise. The technology for timewise integration of signals has been developed. However, acceptance of a wide band width places strong demands on the accuracy of the reflector shape. Investigators in this field quote accuracy requirements of $\lambda/50$, where λ is the shorter wavelength in the sensed band width.

The size of a microwave radiometer is affected by the gain of its reflector inasmuch as the gain must be sufficient to detect weak microwave emissions. The ideal gain G for a diffraction-limited reflector is

$$G = C \left(\frac{D}{\lambda} \right)^2$$

where D = reflector diameter

G = ratio of the power density in the main beam of the reflector to power density if the power were radiating isotropically

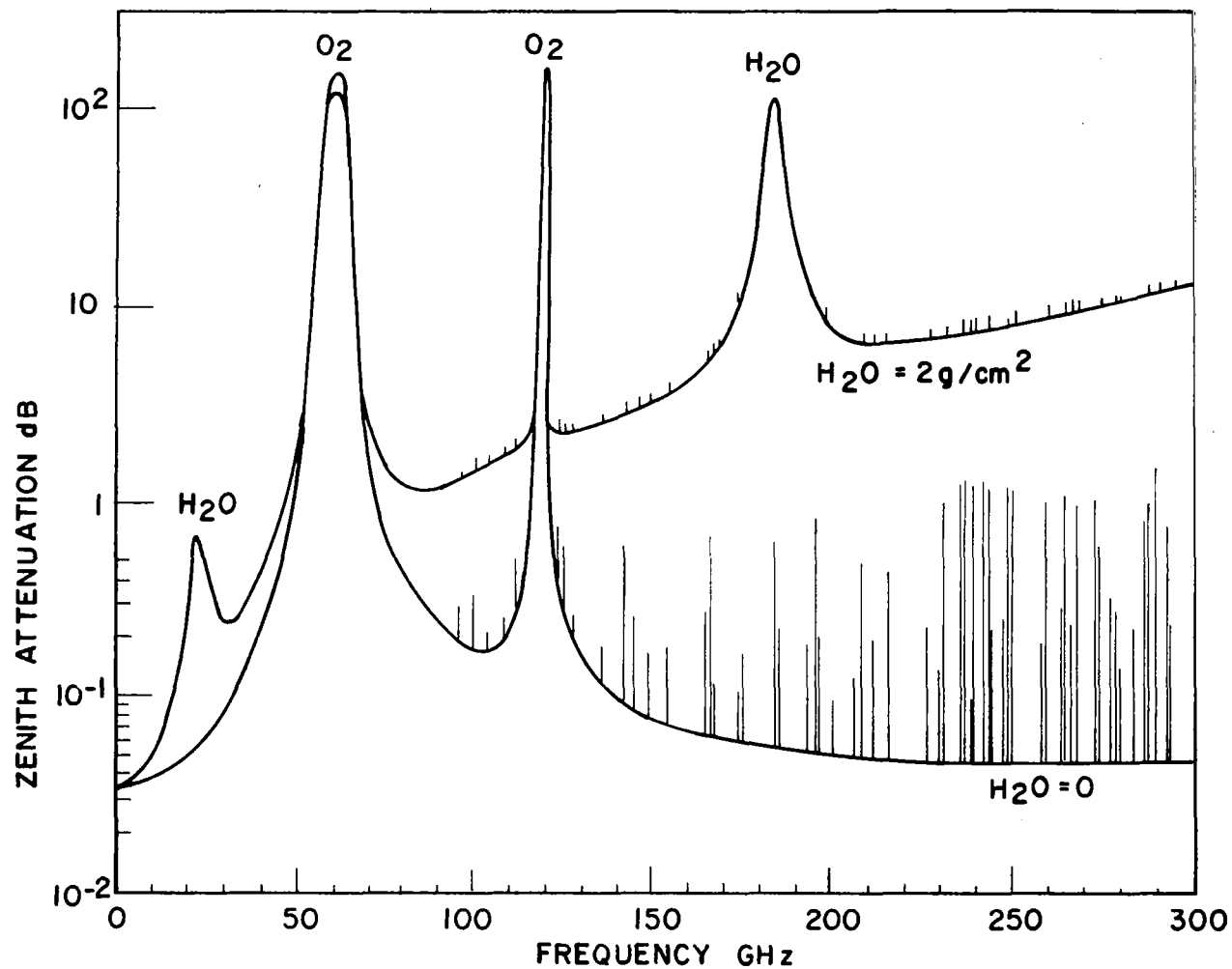


Figure B-1. Zenith Opacity Versus Frequency for Rain and No-Rain Conditions

and $C \approx 10.5$

A second factor affecting the size of a reflector is the required resolution. Resolution size r is approximately related to reflector diameter, wavelength, and range R by the formula

$$r = 1.25 \frac{\lambda}{D} R$$

Resolution is adversely affected by inaccuracies in reflector shape; accordingly, inaccuracies must be minimized where high resolution is required. To avoid significant degradation, inaccuracies should be limited to less than $\lambda/50$.

Thus, the sensitivity limits of radiometers and the need for small resolution sizes will result in demands for large and accurately shaped reflectors.

Following is a list of several potential applications for microwave radiometers, their resolution requirements, their typical wavelengths, and comments on each application.

Microwave Radiometry Applications

Soil Moisture

Observation wavelength = 3.3 to 30 cm

Required resolution = 200 m

Comments: Because of vastly different dielectric constants between soil and water, radiometers can detect moisture in soil. Wavelengths of about 3.3 cm can be used to measure moisture down to a few centimeters below the surface, while 30-cm wavelengths can penetrate tens of centimeters. Over land areas the microwave emission is affected by soil moisture, surface roughness, and surface temperature. When soil moisture is greater than about 10 to 15 percent by weight, the microwave emission decreases almost linearly with moisture content.

Dynamic Properties of Ocean Surface

Observation wavelength = 1.5, 3.6, and 6 cm

Required resolution = 100 to 150 km

Comments: Sensing the ocean surface is of considerable importance to meteorology and oceanography. Remote sensing of surface conditions from satellites offers enormous potentiality because much of the ocean cannot be monitored regularly and adequately by surface

or aircraft-mounted sensors. Both surface roughness and foam tend to increase the brightness temperature (emissivity) of the sea surface with approximately equal effects at 3.6- and 1.5-cm wavelengths. The exact wavelength is not critical as long as surface-temperature effects and atmospheric effects can be separated. The effect of surface temperature has been calculated to be minimal from 1.5 cm to about 15 cm.

Ocean Surface Temperature

Observation wavelength = 5 to 6 cm

Required resolution = 5 to 150 km

Comments: Microwave emission due to sea-surface temperature has a broad maximum centered at wavelengths of approximately 6 cm.

Salinity

Observation wavelength = 21 to 30 cm

Required resolution = 0.1 to 5 km

Comments: At wavelengths of about 30 cm the salinity of water significantly affects surface emission or brightness temperature. Measurements at 21 cm near the mouth of the Mississippi River generally substantiate this salinity dependence.

Sea Ice Detection

Observation wavelength = 0.8 to 10 cm

Required resolution = 200 m

Comments: Open water and ice are easily distinguishable by microwave radiometry. Shorter wavelengths should be observed since they give better resolution for a constant antenna size.

Location and Mapping of Storm Cells

Observation wavelength = 0.8 to 1.35 cm

Required resolution = 1 to 2 km

Comments: The Nimbus 5 microwave spectrometer (see Ref. B-1) monitored atmospheric water vapor and rain respectively at 1.35 cm, the waterline, and at 0.96 cm in the window between H₂O and O₂ absorption (see Fig. B-1). Some information on the altitude distribution of water vapor can be obtained with an additional channel on the edge of the H₂O line at approximately 1.4 cm. Channels in the 0.3-cm, 0.2-cm, and shorter-wavelength atmospheric windows should be useful for measuring smaller amounts of rain and ice, and for information on the drop-size distribution. The shorter-wavelength channels would also be useful for monitoring

atmospheric water over land surfaces, where the surface emission dominates measurements at longer wavelengths.

Figure 8 in the main text shows, for observations from both low and synchronous altitudes, the reflector diameters required for the applications of the above list. When the diameters and operating frequencies shown in Figure 8 are compared with existing antenna technology (see Fig. 1 in the main text), it is seen that some of the indicated observations are possible using present antennas because their large resolutions are acceptable. However, agricultural and geological observations generally require resolution sizes smaller than 500 m (often as small as 50 m), and these resolutions require larger reflectors than those used on present antennas.

The demand for accuracy of reflector shape does not diminish for reflectors of large diameter. For example, at 10 GHz (3-cm wavelength) the rms surface tolerance must still be about $\lambda/50$ or 0.06 cm, an exceedingly small tolerance for a space-based structure whose diameter might need to be 5 to 10 km. Therefore, structural technology for microwave radiometry must be significantly increased to produce reflectors which are large, accurately shaped, and capable of being deployed or assembled in space.

Note that Reference B-2 contains an extensive bibliography for microwave radiometry.

APPENDIX B REFERENCES

- B-1. Staelin, D. H.; Barrett, A. H.; et al: Microwave Spectrometer on the Nimbus 5 Satellite: Meteorological and Geophysical Data. Science, Vol. 182, December 1973, pp. 1339-1341.
- B-2. Tomiyasu, K.: Remote Sensing of the Earth. Proceedings of the IEEE, January 1974, pp. 86-92.